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RESEARCH MEMORANDUM

INVESTIGATION OF FLAP-TYPE AILERONS ON AN UNTAPERED WING HAVING AN ASPECT RATIO OF 3.7, 45° SWEEPBACK, AND AN NACA 65A009 AIRFOIL SECTION

TRANSONIC-BUMP METHOD

By Richard G. MacLeod

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NATIONAL ADVISORY COMMITTEE FOR AERONAUTICS

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SUMMARY

An investigation to determine the lateral control characteristics of a 20-percent-chord flap-type aileron of various spans on a semispan wing-fuselage model was made in the transonic speed range. The wing of the model had 45° of sweepback, an aspect ratio of 3.7, a taper ratio of 1.0, and an NACA 65A009 airfoil section parallel to the free stream. Rolling moments were obtained through a small range of angles of attack and aileron deflections. Lift data on the complete model are also included.

The experimental results were in good agreement with those predicted from low-speed theory and other experimental data at a Mach number of 0.6, and the relative spanwise effectiveness of the aileron remained fairly constant throughout the Mach number range tested.

INTRODUCTION

One of the problems arising with the use of high-speed aircraft has been that of securing adequate lateral control, particularly in the transonic speed range. Recent investigations with rocket-powered test vehicles, by means of the transonic-bump technique and conventional wind tunnels, have added to the general knowledge of controls, but the actual data which are available are few in comparison with those needed for design purposes. The present investigation which supplies some additional information on the subject was made to determine the effectiveness of flap-type ailerons on an untapered sweptback wing. The

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configurations investigated were a full-span and two semispan ailerons, one located at the outboard and the other at the inboard end of the wing.

The model was tested from a Mach number of 0.60 to 1.15 by means of the transonic-bump technique. The data are presented in the form of rolling-moment coefficients for a small range of angles of attack and aileron deflections.

MODEL AND APPARATUS

The semispan wing used in the investigation had 45° of sweepback, a taper ratio of 1.0, an aspect ratio of 3.7, and an NACA 65A009 airfoil section parallel to the free air stream (fig. 1). The wing was made of steel and the fuselage was made of brass with all surfaces polished. The wing was mounted in the center of the fuselage vertically and had no dihedral or incidence. The fuselage was a cylindrical body with an ogive nose and was shaped to the contour of the bump (fig. 2). A $\frac{1}{8}$-inch plate was fastened to the fuselage in order to raise the fuselage-wing intersection to the root end of the inboard flap and still permit the use of an available fuselage.

The flaps were made integral with the wing by cutting grooves 0.03 inch wide along the 80-percent-chord line on the upper and lower surfaces of the wing (fig. 1). After setting the control at the desired deflection by bending the metal along the grooves, the grooves were faired with wax.

The model was mounted on an electrical strain-gage balance wired to calibrated galvanometers in order to measure the aerodynamic forces and moments. The balance was mounted in a chamber within the bump, and the chamber was sealed except for a small rectangular hole through which an extension of the wing passed. This hole was covered by a $\frac{1}{32}$-inch end plate located approximately 0.03 inch above the bump surface.

COEFFICIENTS AND SYMBOLS

C_L lift coefficient $\left(\frac{\text{Twice lift of semispan model}}{qS}\right)$

C_l rolling-moment coefficient at plane of symmetry $\left(\frac{\text{Rolling moment of semispan model}}{qSb}\right)$

C_{l_a} rolling-moment coefficient produced by the control (rolling-moment coefficient of the entire wing with control deflected minus rolling-moment coefficient of the entire wing with undeflected control)

q effective dynamic pressure over span of model, pounds per square foot $\left(\frac{1}{2}\rho V^2\right)$

S twice wing area of semispan model, 0.116 square foot

b twice span of semispan model, 0.654 foot

$\bar{c}$ mean aerodynamic chord of wing, 0.177 foot

c local wing chord, feet

y spanwise distance from plane of symmetry, feet

y_i spanwise distance from plane of symmetry to inboard end of control, feet

ρ mass density of air, slugs per cubic foot

V free-stream air velocity, feet per second

M effective Mach number over span of model

M_a average chordwise local Mach number

M_l local Mach number

R Reynolds number of wing based on $\bar{c}$

α angle of attack, degrees referred to wing root chord line

δ control deflection relative to wing-chord plane measured perpendicular to control hinge axis, degrees

b_a control span measured perpendicular to plane of symmetry, feet

$$C_{L_\alpha} = \left(\frac{\partial C_L}{\partial \alpha}\right)_\delta$$

$$C_{l_\delta} = \left(\frac{\partial C_l}{\partial \delta}\right)_\alpha$$

Subscripts:

c corrected

u uncorrected

CORRECTIONS

The rolling-effectiveness parameters presented herein represent the aerodynamic effects on a complete wing produced by the deflection of the aileron on only one semispan of the complete wing. Reflection-plane corrections have been applied to the data throughout the Mach range tested. The correction factors which were applied to the parameters are given in figure 3. The values of the correction factors given in figure 3 were obtained from unpublished experimental low-speed data and theoretical considerations. Unpublished results of high-speed tests of a similar model mounted on a sting support indicate that the results obtained by applying the low-speed corrections give a better representation of true conditions at high Mach numbers than uncorrected data.

No attempt has been made to correct the rolling-moment data for increments of rolling moment due to lift increase on the wing-fuselage end plate (fig. 1) produced by control-surface deflection. From unpublished data, this effect has been found to be of little significance for either inboard or outboard control surfaces. The maximum deflection of the tip under aileron load was found to be 0.32°; this effect was considered to be within the accuracy of the data.

TESTS

The tests were conducted in the Langley high-speed 7- by 10-foot tunnel using an adaptation of the NACA wing-flow technique for obtaining transonic speeds. The technique used involves placing the model in the high-velocity flow field generated over the curved surface of a bump on the tunnel floor (reference 1). Typical contours of local Mach number in the vicinity of the model location on the bump with model removed are shown in figure 4. No attempt has been made to evaluate the effects of the chordwise and spanwise Mach number variation. The long dashed lines near the root of the wing in figure 4 indicate a local Mach number 5 percent below the maximum value and represent the estimated extent of the bump boundary layer. The effective test Mach number was obtained from contour charts similar to those presented in figure 4 by using the relationship

$$M = \frac{2}{S}\int_{0}^{b/2} cM_a \, dy$$

The variation of the mean test Reynolds number with Mach number is shown in figure 5.

Lift and rolling-moment data were obtained for the model configuration tested through a Mach number range of 0.60 to 1.15, at angles of attack of -2°, 0°, and 2° and in the aileron-deflection range of -5° to 10°.

RESULTS AND DISCUSSION

Lift-coefficient data on the complete model are presented in figure 6. The lift-curve slope (fig. 7) reaches a maximum value of 0.054 at a Mach number of 0.98; the values were below those predicted for the wing alone by theory of reference 2.

The results of the lateral-control investigation on an untapered wing of 45° sweep are presented in figures 8, 9, and 10. Figure 9 indicates a general decrease in aileron effectiveness between the Mach numbers of 0.9 and 1.0 for all three aileron configurations. The data show that the relative spanwise effectiveness of the aileron remains fairly constant throughout the Mach number range tested and is in good agreement with the results of reference 3.

Figure 10 presents a comparison of the experimental values of aileron effectiveness determined by three different methods with the theoretical curve of C_{l_δ} for a rigid wing (reference 4). Identical models were used for the rocket-powered-vehicle test (reference 5) and for the wind-tunnel test (unpublished), whereas the transonic-bump model was of a considerably smaller scale and differed slightly in aileron span. The value of C_{l_p} (0.285) used to attain C_{l_δ} from the values of $pb/2V$ of reference 5 was obtained from the unpublished wind-tunnel tests and is in good agreement with theory (reference 6). The results of the three methods presented compare favorably with each other within the accuracy of the data. On a flexible wing, however, the effectiveness of a given aileron may be materially changed. The wing twist induced by aileron deflection could considerably reduce the effectiveness of controls located at or near the tip and have only small effect on a control located near the root of the wing.

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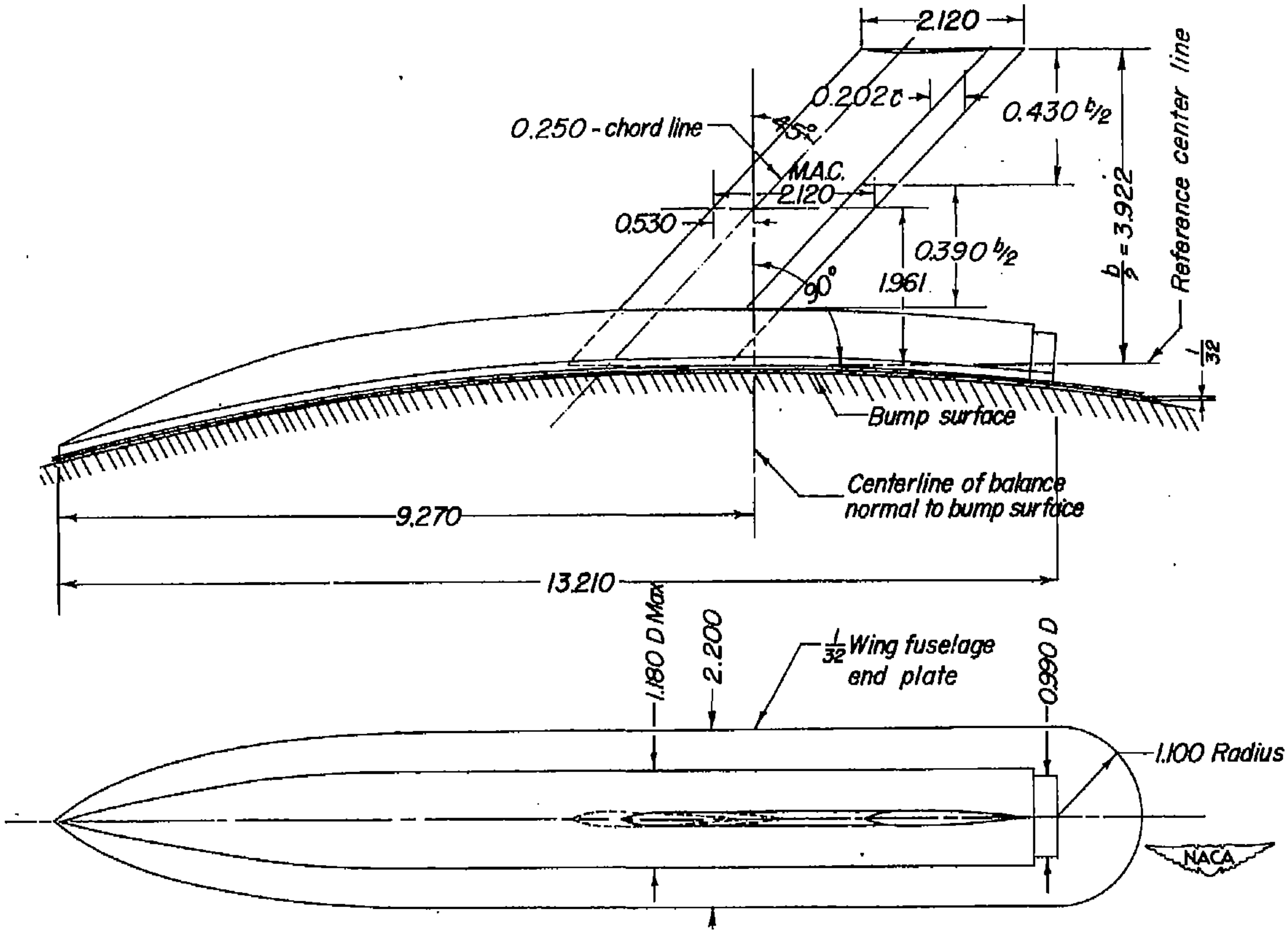
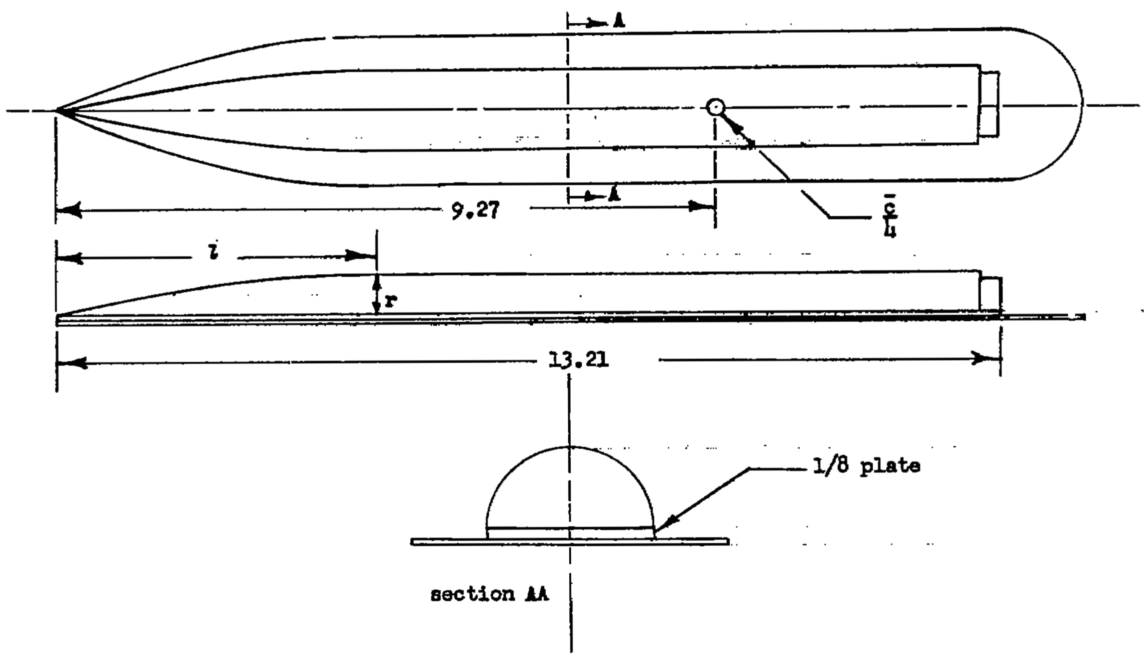


Figure 1.- General arrangement of model with 45° sweptback wing, aspect ratio 3.7, taper ratio 1.0, and NACA 65A009 airfoil. All dimensions are in inches.

Cylindrical body and end plate



Cylindrical body Ordinates, inches	
l	r
0	0
.59	.144
1.18	.271
1.77	.372
2.36	.462
2.95	.533
3.54	.575
4.13	.589
7.00	.589
10.50	.589
12.92	.589
12.92	.495
13.21	.495

Figure 2.- Drawing and ordinates of the cylindrical body. (All dimensions are in inches.)

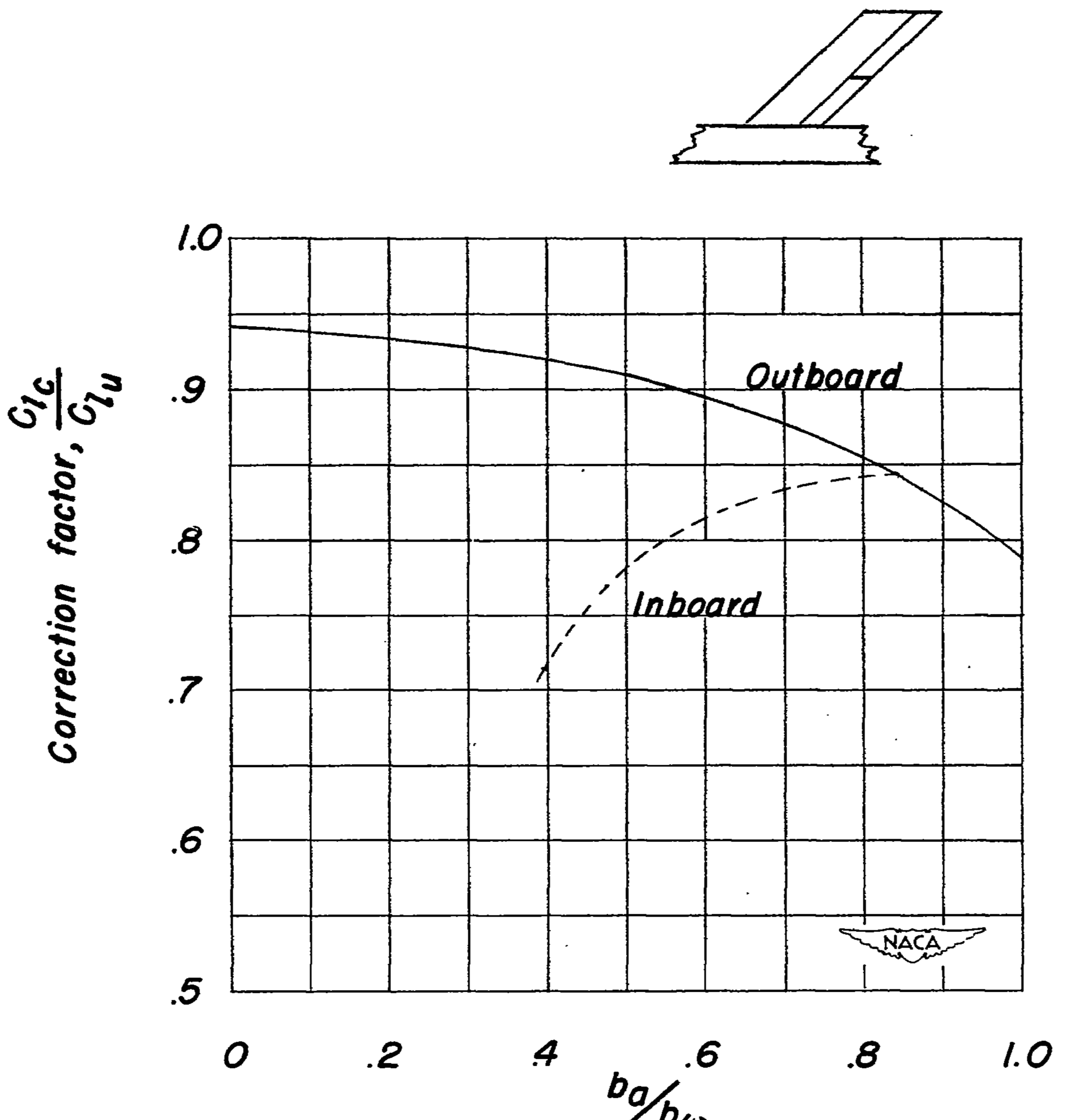


Figure 3.- Reflection-plane correction factors for inboard and outboard controls of various spans for a wing of 45° of sweepback, aspect ratio 3.7, taper ratio of 1.0, and NACA 65A009 section.

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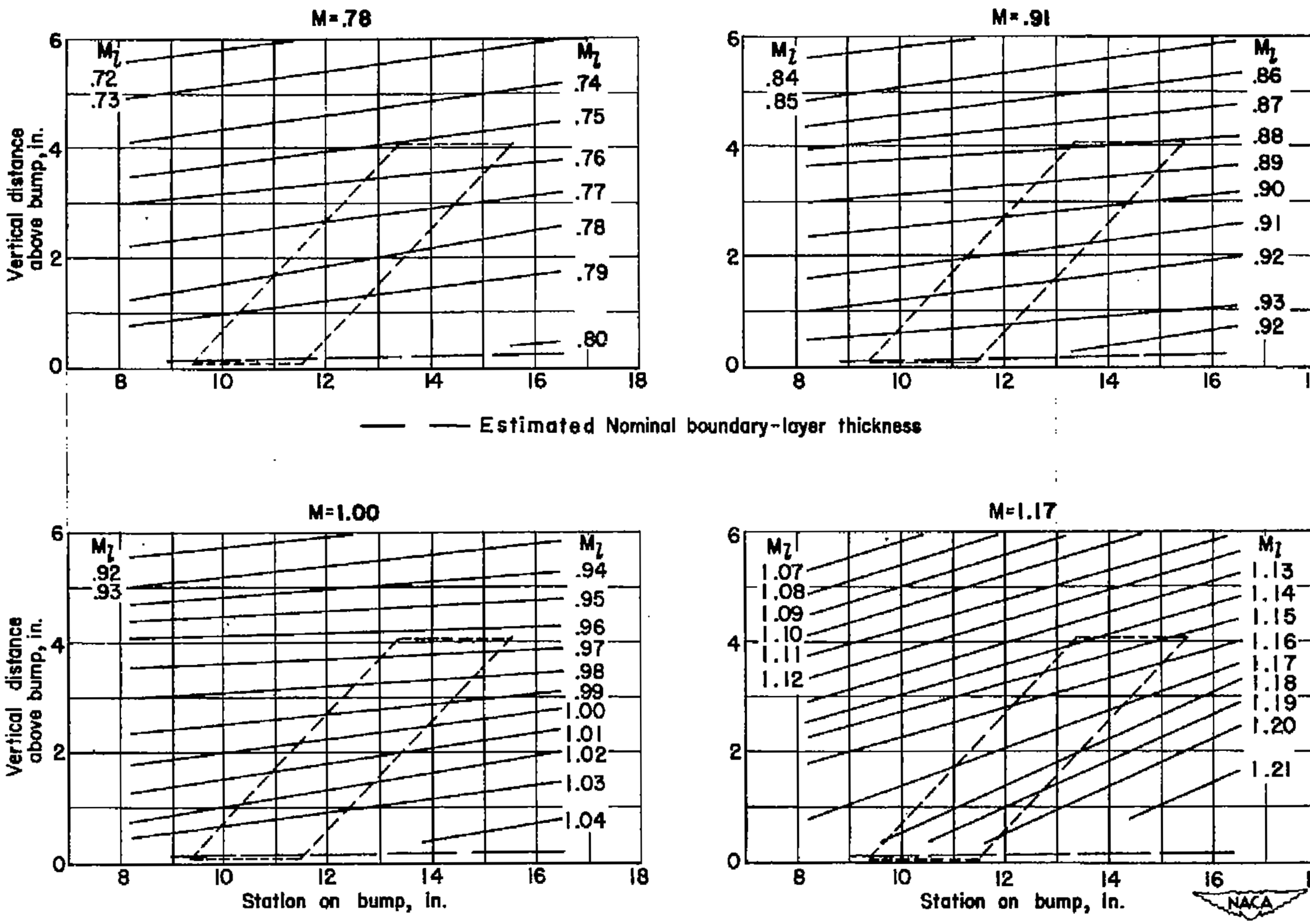

Figure 4.- Typical Mach number contours over transonic bump in region of model location.

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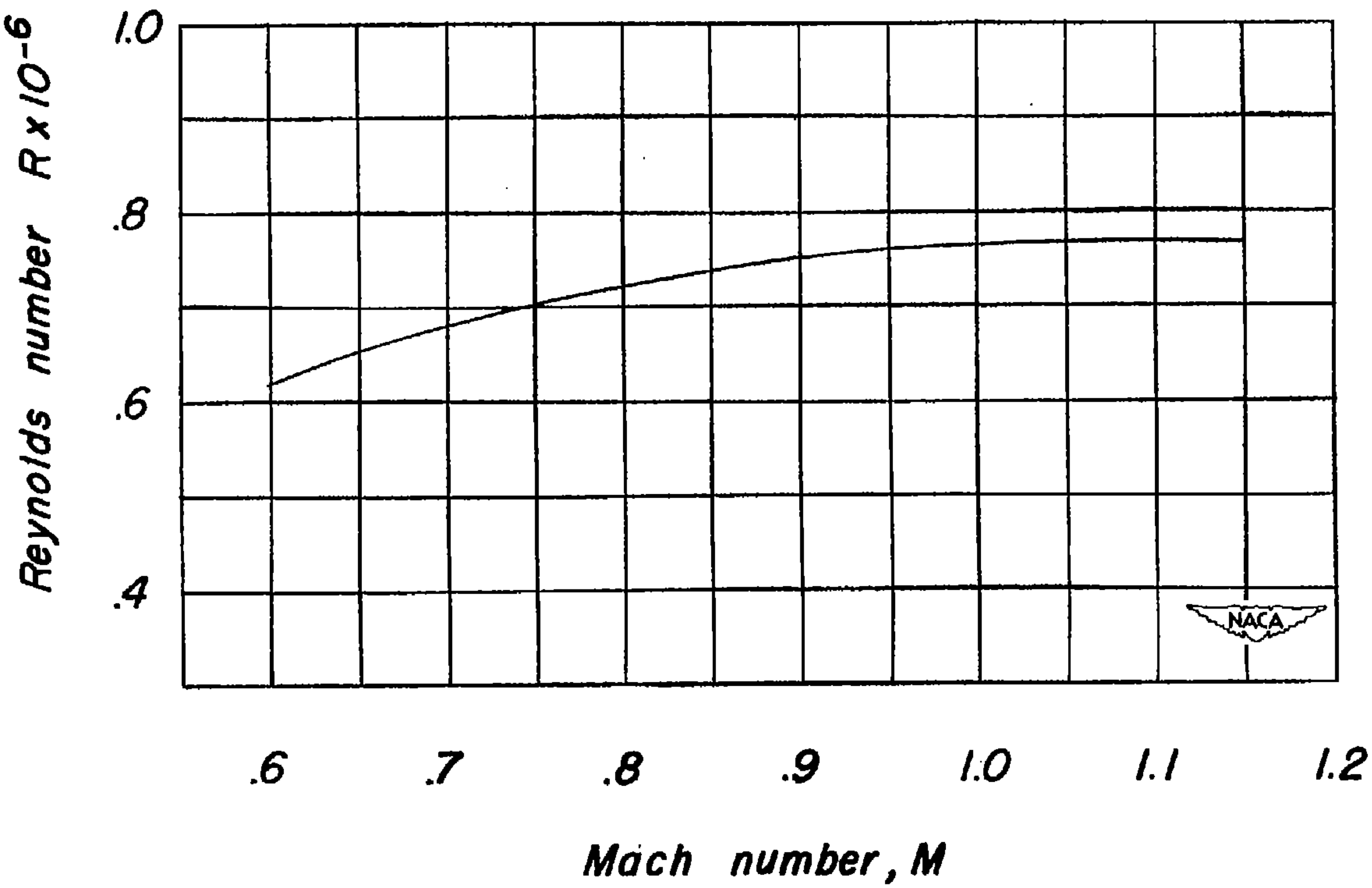


Figure 5.- Variation of test Reynolds number with Mach number for model with 45° sweptback wing, aspect ratio 3.7, taper ratio 1.0, and NACA 65A009 airfoil.

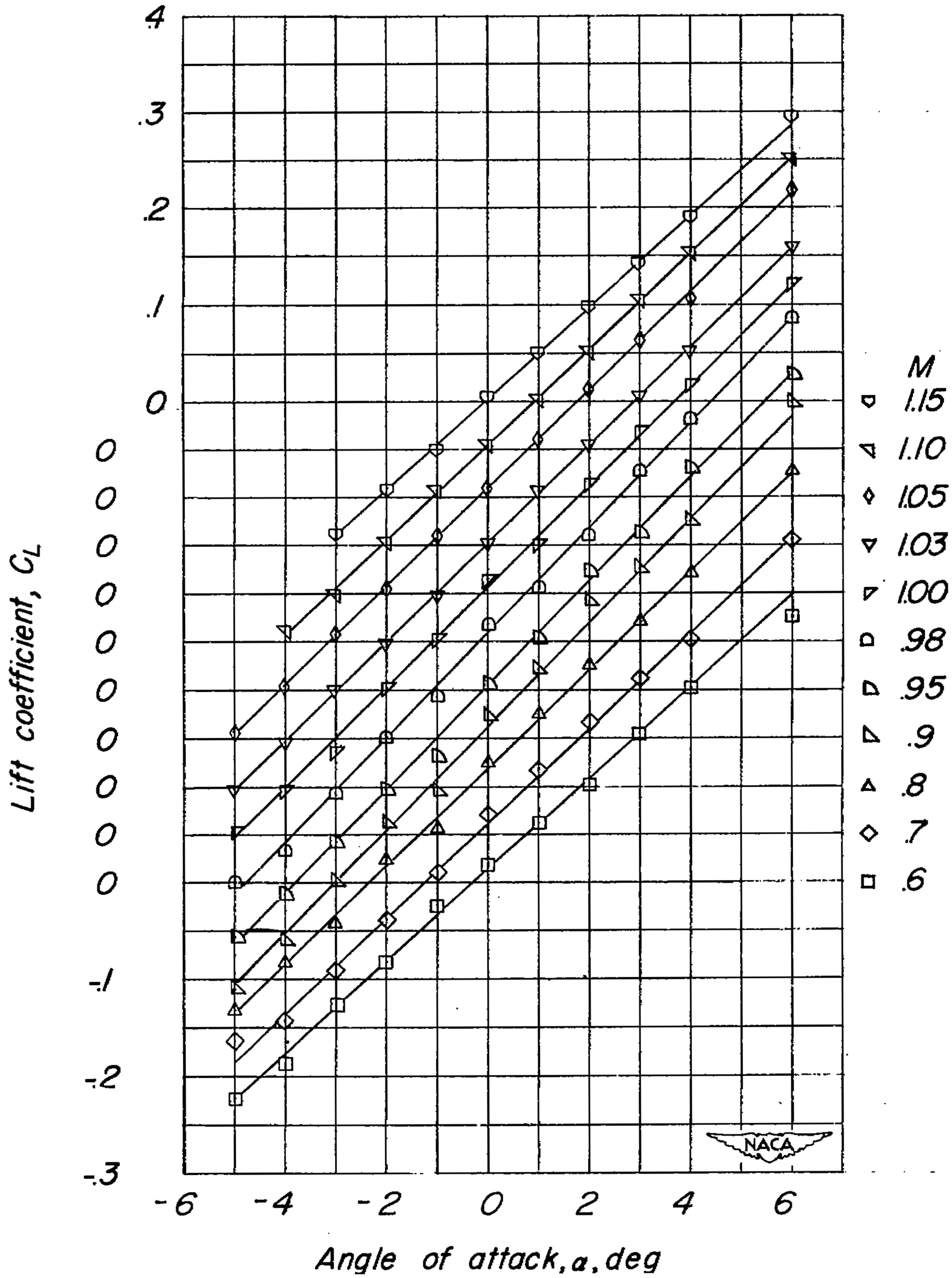


Figure 6.- Variation of lift coefficient with angle of attack for various Mach numbers.

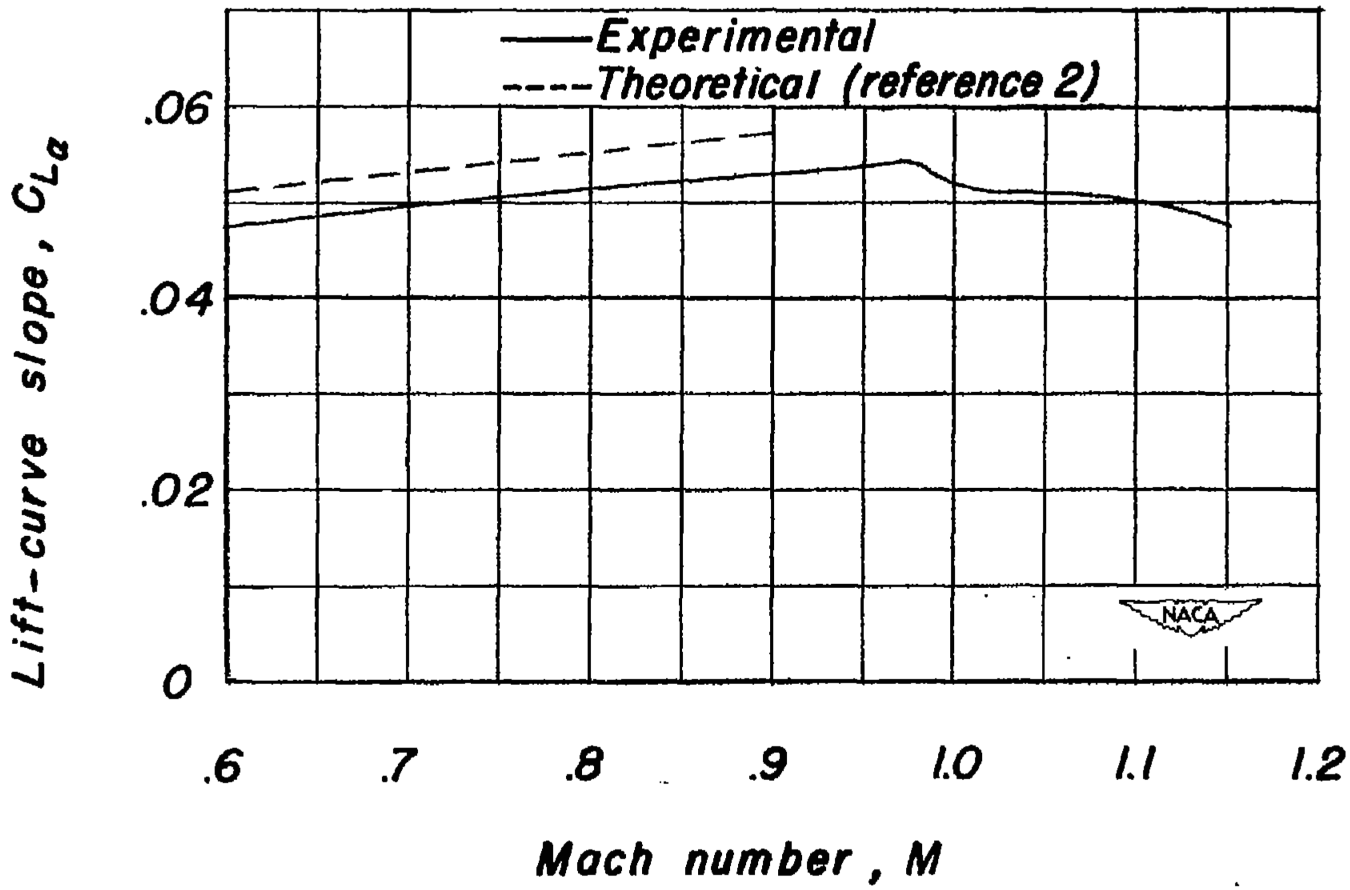
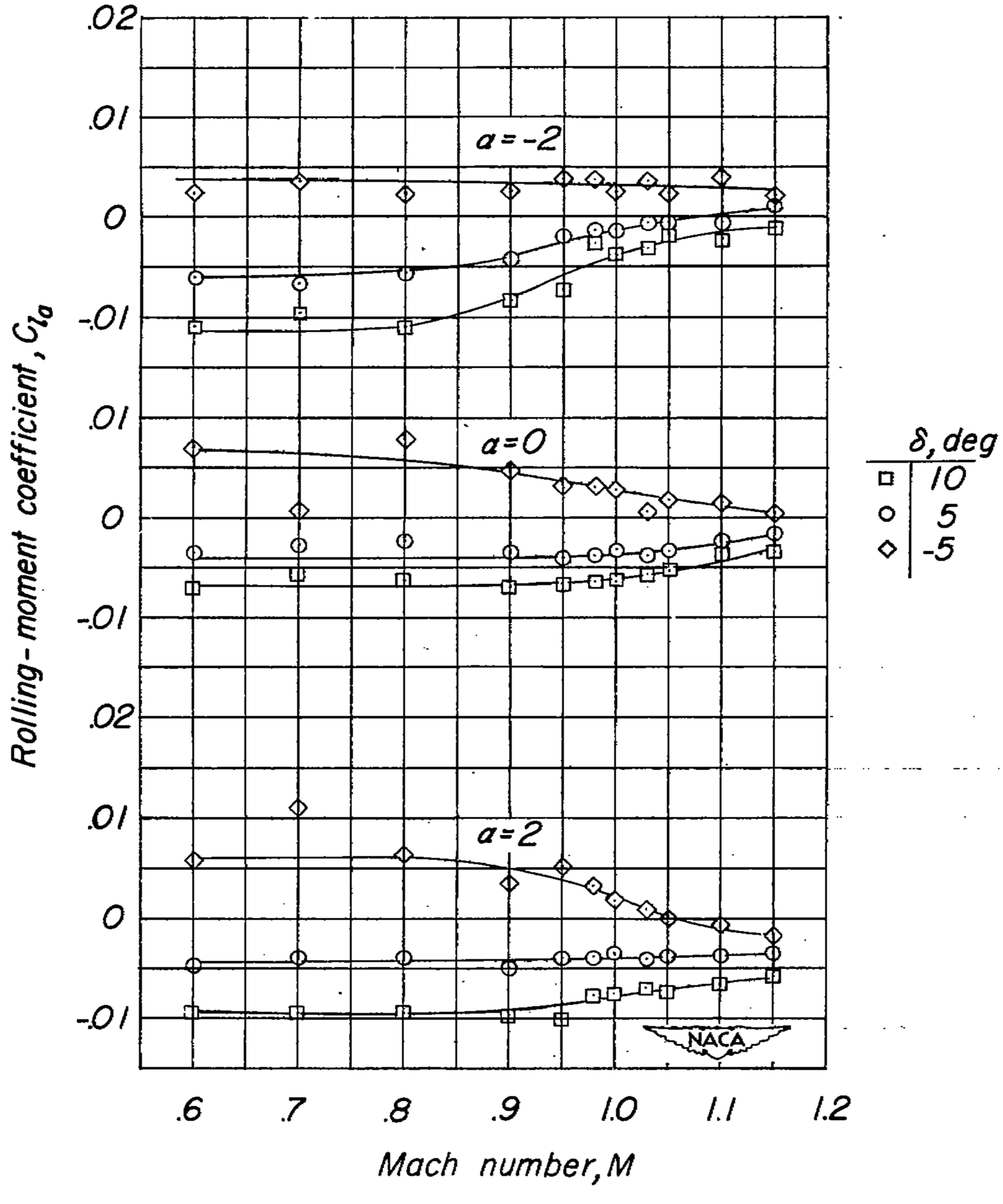


Figure 7.- Variation of lift-curve slope with Mach number.



(a) Outboard flap.

Figure 8.- Variation of rolling-moment coefficient with Mach number for various control deflections.

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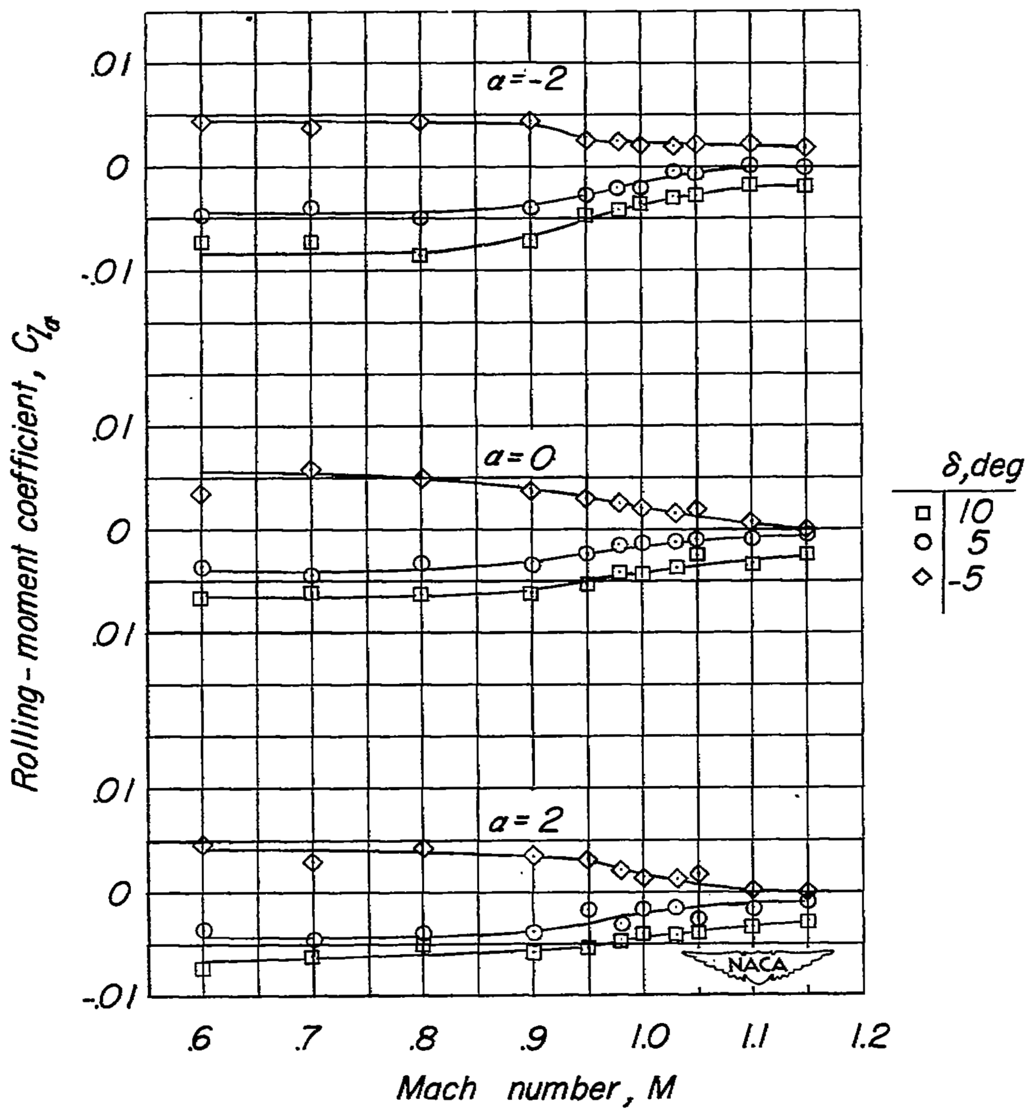
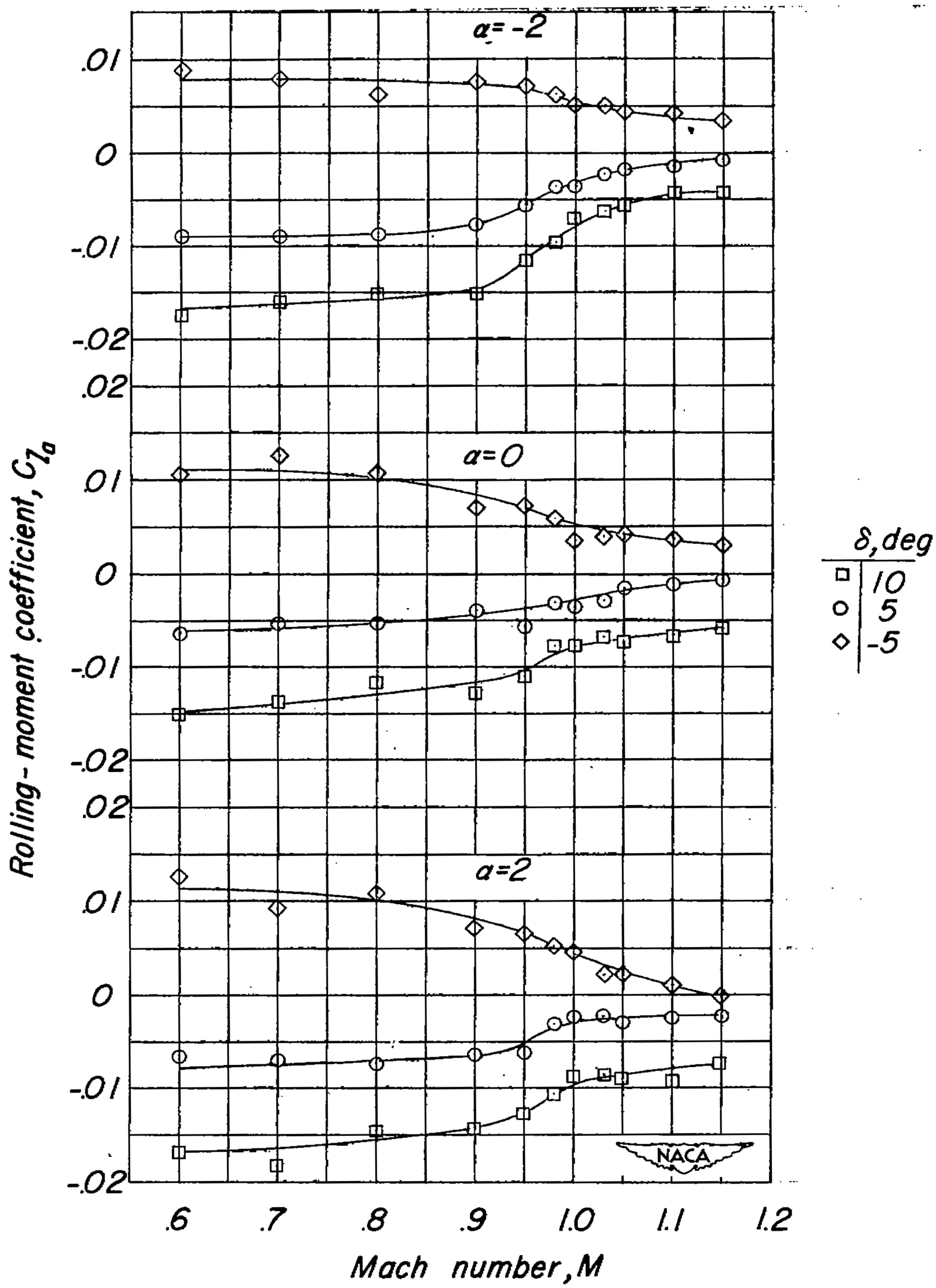


(b) Inboard flap.

Figure 8.- Continued.



(c) Full-span flap.

Figure 8.- Concluded.

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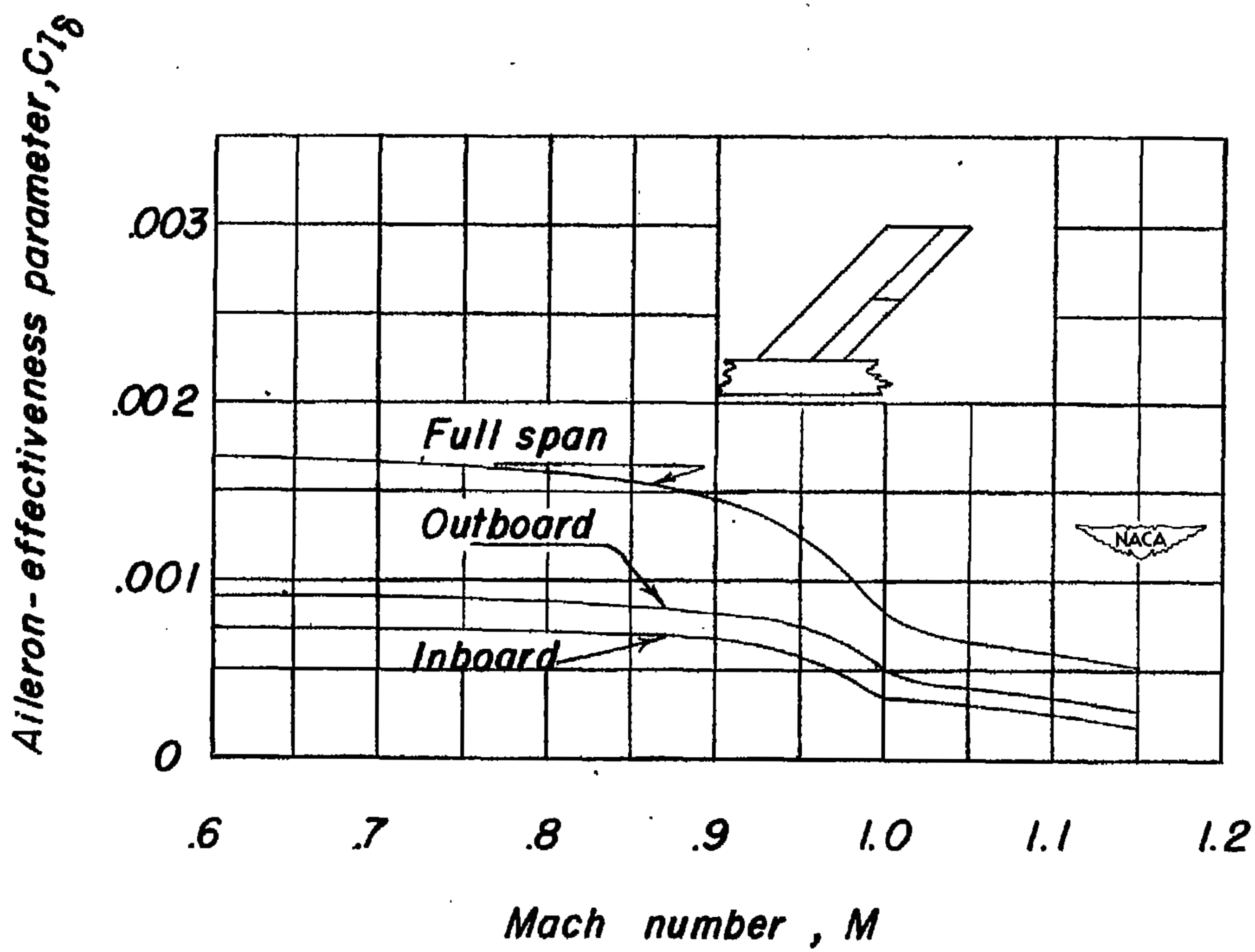

Figure 9.- Variation of aileron-effectiveness parameter with Mach number. $\alpha = 0^{\circ}$.

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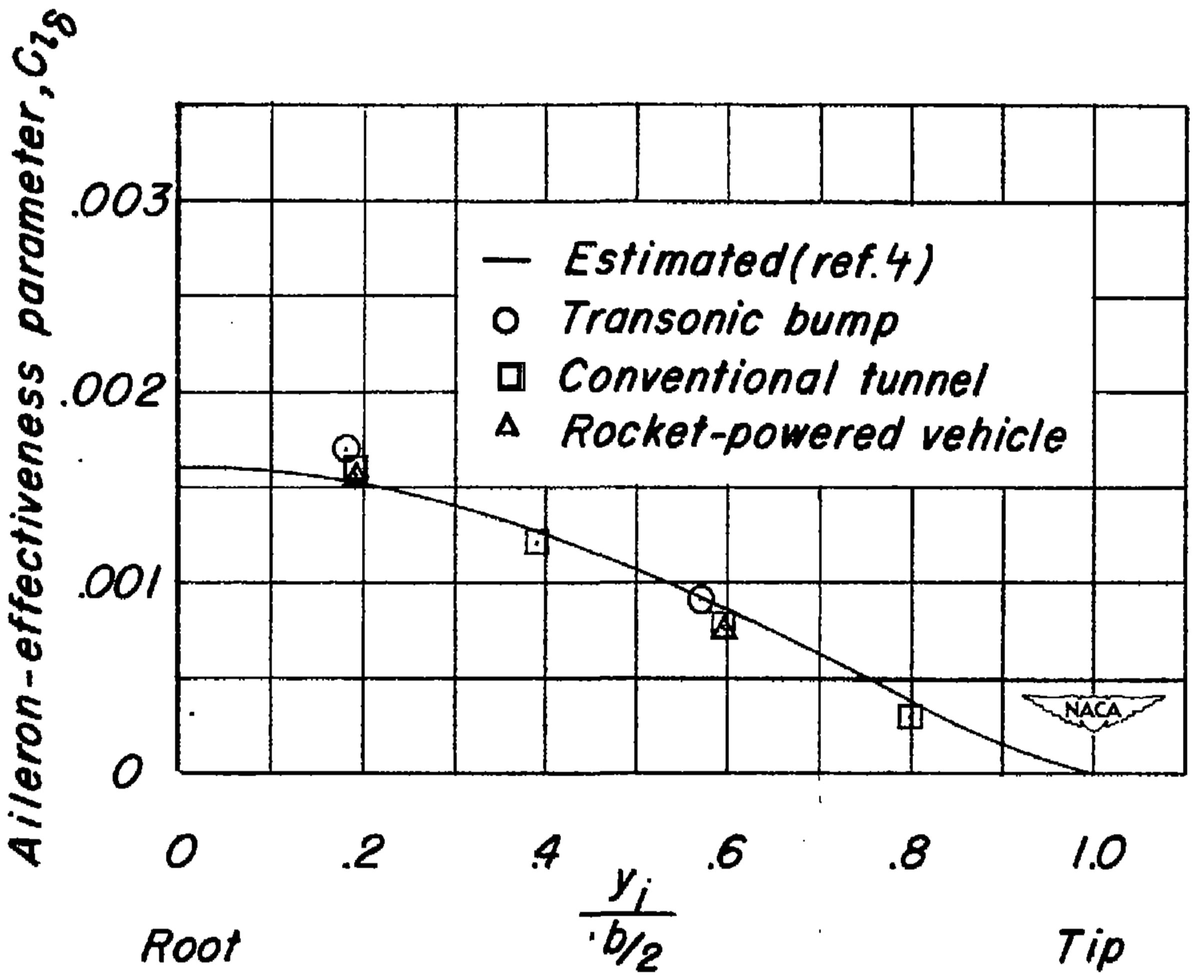

Figure 10.- Comparison of the experimental and estimated variation of aileron effectiveness with control span. $\alpha = 0^{\circ}$. $M = 0.6$.

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